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Dr. James Greer	DFEM/CAStLE	333-3618	lgm	20130419

SUBJECT

Clearance for Material for Public Release USAFA-DF-PA- 295

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SUMMARY

1. PURPOSE. To provide security and policy review on the document at Tab 1 prior to release to the public.

2. BACKGROUND.
Authors: Sandeep R. Shah, Todd B. St John, Ian L. Pryce, Jr., Nex One, Inc., Gregory A. Shoales and James M. Greer, Jr, Center for Aircraft Structural Life Extension

Title: Multi-Site Fatigue Testing and Characterization of Fuselage Panels from Aging Aircraft Structure

Circle one: Abstract Tech Report Journal Article Speech Paper Presentation Poster Thesis/Dissertation Book Other: CONFERENCE PROCEEDINGS

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[] CRADA (Cooperative Research and Development Agreement) exists
[] Photo/ Video Opportunities [] STEM-outreach Related [] New Invention/ Discovery/ Patent

Description: Paper is to be presented at the International Committee on Aeronautical Fatigue and Structural Integrity Conference 2013, Jerusalem, Israel 5-7 June 2013.

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3. DISCUSSION: This research was performed under the direction of the Center for Aircraft Structural Life Extension.

4. RECOMMENDATION: Department Head or designee review as subject matter expert. DFER review for policy and security and provide public release clearance.

James M Greer, Jr.
JAMES M. GREER, Jr., PhD, PE
Technical Director, CAStLE

1 Tab
1. Conference Paper

Multi-Site Fatigue Testing and Characterization of Fuselage Panels from Aging Aircraft Structure

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Abstract: Multi-site fatigue damage is a common problem in the riveted lap joint structure of aging aircraft. Modeling and characterization of such damage is an especially daunting task. In this effort we present the results from fatigue tests which were performed on fuselage lap joints extracted from various retired USAF military aircraft. The test specimens were extracted from a variety of fuselage locations of these aircraft and subjected to spectrum loading simulating fuselage pressurization cycles. The test panels varied in width from 15 to 22 inches and typically had three (3) rows of fasteners in the lap joint. Some spot welded lap joint panels were also tested during the larger program; however, only the results from mechanically fastened joints are presented here. The results from approximately 25 panels from two different aircraft are presented here. Also presented is the process used to prepare the panel test specimens. Instrumentation consisting of strain gages on either side of the lap joint was used to verify symmetric load introduction across the width of the specimen. All tests were run to lap joint failure. Post failure, the fatigue damage at each fastener location of the lap joint was characterized in detail. Wherever possible the first fastener hole to exhibit fatigue cracking was identified. The data are presented in graphical as well as tabular format for easy incorporation into validation and development testing for modeling software platforms such as AFGROW and NASGRO. These data also provide an excellent map for damage initiation and propagation, which could be used to validate modeling and simulation efforts aimed at predicting the life and reliability of similar aircraft structure.

INTRODUCTION

The United States Air Force fixed wing aircraft fleet is maintained primarily using the damage tolerance philosophy [1]. This philosophy assumes the existence of defects and flaws in structures, and the design and maintenance of the fleet is based on the tolerance of such damage in service. The application of damage tolerance analysis requires evaluating the residual strength or life of the structure in the presence of such defects. Various fatigue crack growth models can be used to determine the life of a structure once the flaw or defect is detected. However, the analysis quickly becomes complicated when a number of damage sites are in close vicinity and the load transfer and load shedding to the neighbouring structures are dependent on the crack sizes at each location.

Multi-site damage (MSD) is the occurrence of multiple cracks at multiple locations in close vicinity. This is typically true in the case of riveted lap joints, where a number of holes along the fastener lines can act as crack initiation points. The semi-monocoque fuselage structure has a skin shell with riveted lap joints reinforced by stringers and frames. The riveted skin shell structure is stressed primarily by the pressurization hoop stress during flight. Understanding the residual strength and/or residual life of lap joints in fuselage skin having multiple cracks at rivet holes is extremely important to assessing structural health. Once MSD cracks link up, the strength and the life of the structure are dramatically reduced.

MSD in aircraft fuselage structure gained attention after the in-flight failure of an Aloha Airlines Boeing 737 fuselage lap joint in 1988. In this incident, small cracks emanating from the rivet holes linked up and led to catastrophic failure of the fuselage skin in flight. Various deterministic and probabilistic prediction models have been developed to predict the failure due to MSD. [2,3,4]. The success of deterministic or probabilistic models depends upon the data input provided and the availability of results to calibrate the models. Most of the data inputs have significant variability and limited availability. An effective approach in this case can be Monte Carlo simulation, in which the variability in data input has less effect the output. (However, more data points improve the calibration of the Monte Carlo simulation.)

There are very few experimental results available to calibrate the simulation models. Most of the experimental data available are from laboratory-assembled lap joints. To our knowledge, significant fatigue failure data from in-service aircraft lap joints are not available. Also, the variability in factory-assembled structure, which is later maintained and/or repaired in field, is much higher than one can expect from laboratory-assembled lap joints. Furthermore, when the specimens are extracted from actual aircraft, the panels have residual stresses due to extant supporting structure. These differences in construction lead to differences in results. In the present work, we have tested and characterized lap joints extracted from actual aircraft. These data could useful inputs for predictive models.

EXPERIMENTAL

The panels for the resent task were extracted from two transport aircraft. The panels were selected from various sections of the fuselage. Typically extracted sections were about 750 mm (30 in) wide and 1500 mm (60 in) tall. Extracted panels initially had other structural elements attached to them. Figure 1 shows one of the as-received test sections with stringers and frames attached to it. Frames were removed and the panels were machined into an hour glass planform. Holes were cut to affix the specimen to the grips for mounting in the test frame as shown in Figure 2. The width of the lap joints after machining varied between 375 mm and 550 mm (15–22 in). The specimen orientation was such that the stringers ran parallel to the loading axis. The applied loading, therefore, represents the hoop stress acting on the fuselage skin. In semi-monocoque structure like this, the bending forces are primarily taken by the stringers and frames and the skin will primarily experience hoop stresses. Thus, the uniaxial testing of these test specimens closely represents the loading experienced by the lap joint during flight.

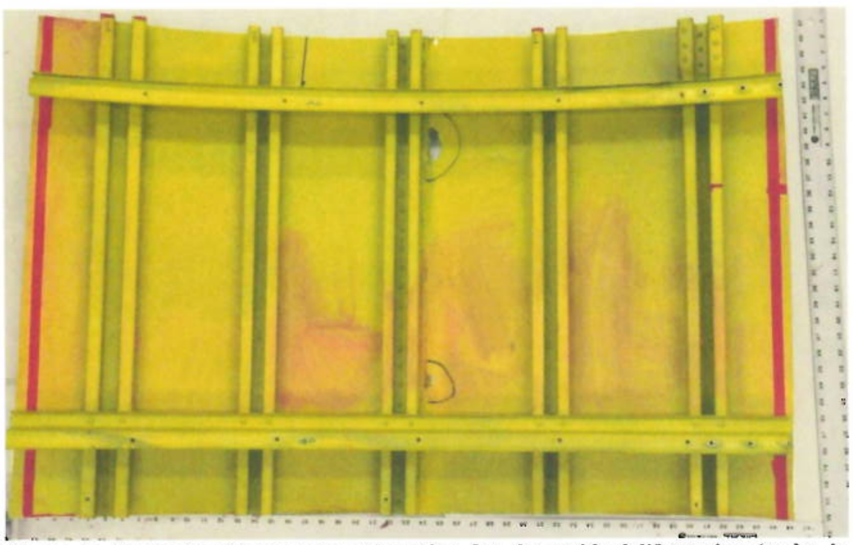

Figure 1: Photo showing as-removed section for the residual life testing (scales in inches).

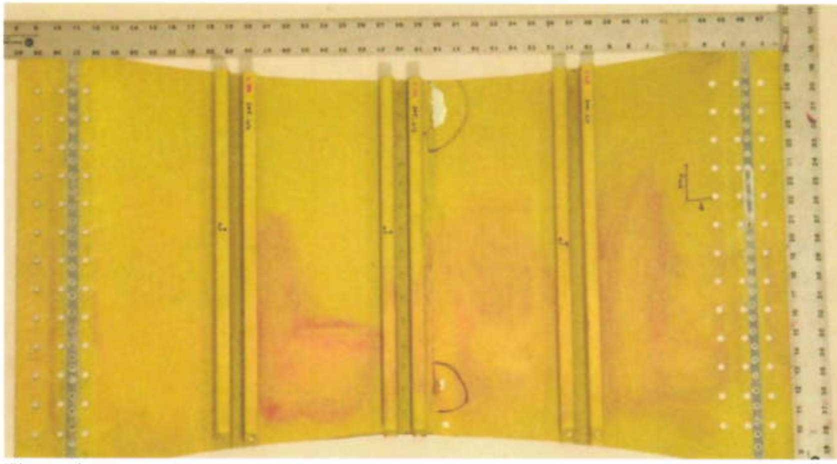

Figure 2: Photo showing the prepared test specimen after machining from the received section of Figure 1 (scales in inches).

The fasteners at the edges of the panel were replaced with next-oversize fasteners with a slight interference fit to prevent crack initiation from edge fasteners. The machined specimen was then instrumented with strain gages. Five strain gages were applied just above the lap joint and five were applied just below the lap joint. They were equally spaced along the width of the specimen and measured the strain in the two skin panels of the lap joint. The specimen was then gripped in steel grips with 3/8 in SAE grade 8 bolts and nuts. These steel grips gradually tapered to 125 mm (5 in) or 150 mm (6 in) tabs, which were gripped in either 500,000 N (110 kip) or 1,000,000 N (220 kip) capacity servo-hydraulic test frame (MTS – Eden Prairie, MN, USA) respectively. The test specimen assembled in the grips was then mounted in the test frame, with tabs of the grips held in the hydraulic wedge assembly of the test frame as shown in Figure 3. The strain gages were monitored using a separate data acquisition computer running LabView from National Instruments (Austin, TX). The load and displacement of the ram were monitored by the control computer of the test frame. The proportional analog output was taken from the machine's controller and fed to the LabView computer for storage.

After the specimen was mounted in the test frame, the data acquisition system was initialized with the panel in a no-load condition. The specimen was then loaded to 4,500 N (1,000 lbf). At this point, strain gage readings were used to verify correct load introduction along the width of the specimen. If out of plane twist was indicated by the strain gage readings, the test machine ram was rotated to eliminate the twisting of the panel. If gages indicated width-wise asymmetric loading, the specimen was released from the hydraulic grips and manually realigned until the strain gages indicated satisfactory loading. This was a significant task for each panel, as the whole assembly wighed close to 1000 N (220 lbf).

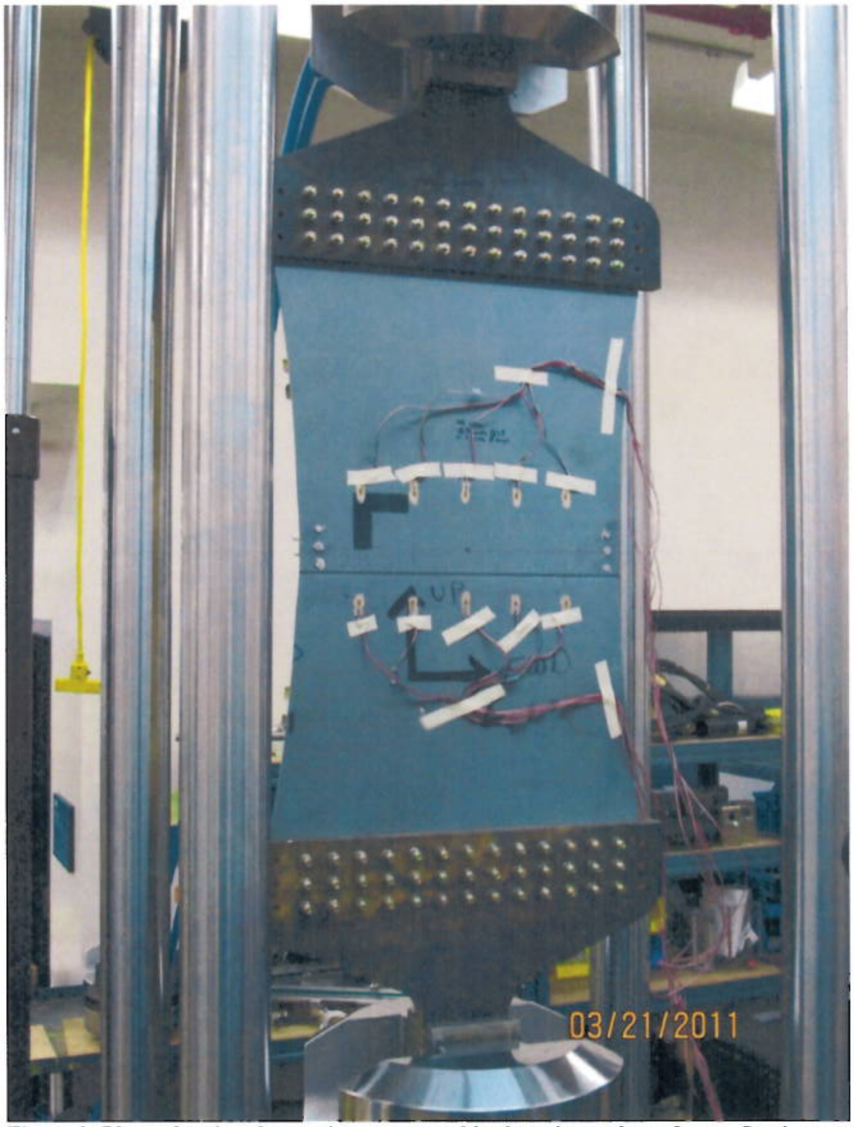

Figure 3: Photo showing the specimen mounted in the grips and test frame. Strain gage locations and replaced edge rivets are visible in the photo.

The testing of the specimen was done to simulate the ground-air-ground (GAG) cycles for a typical mission mix. It was decided to run a marker band spectrum in conjunction with the GAG spectrum. This way, pre-existing crack surfaces in the specimen would be marked for identification. Recall that these panels came from retired aircraft, hence the fuselage panels have seen significant in-service GAG cycles. If pre-existing cracks were interpreted as new cracks, testing would indicate an erroneous number of cycles for the measured crack length. The Marker ban cycle (as plotted in AFGROW software) is shown in Figure 4. The total marker band spectrum is of 8170 cycles, where each marker band is separated by 2000 cycles and each step of the marker band has 100 cycles of 80% load, interspersed with 10 cycles of 100% loads.

The maximum level of the marker band is set at one third the maximum level of the actual GAG cycle loading.

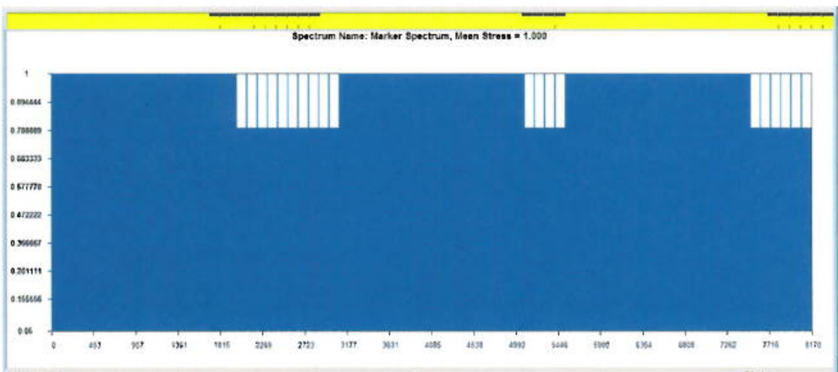

Figure 4: A plot showing a typical marker band spectrum as plotted in AFGROW. This spectrum consists of three marks of 10, 4 and 6 at 80% maximum load level.

The ground air ground cycle (GAG) spectrum was provided by the OEM based on a typical mission mix of the fleet from which the aircraft was retired. The data were provided in terms of the pressure inside the fuselage, which was converted to the hoop stress depending on the location of the panel within the aircraft. The normalized load levels are shown in Figure 5. There are variations in stress levels depending on the nature of the particular mission. The block shown in Figure 5 is for 100 missions or 100 GAG cycles. The test machine considers each local max-min level as one cycle, so to interpret the results correctly, based on this spectrum, we have to normalize 108 cycles counted by the machine as 100 cycles. In calculating the final life of the specimen, it was necessary to subtract the 8170 marker band cycles and then normalize the total number of cycles by the factor of 1.08.

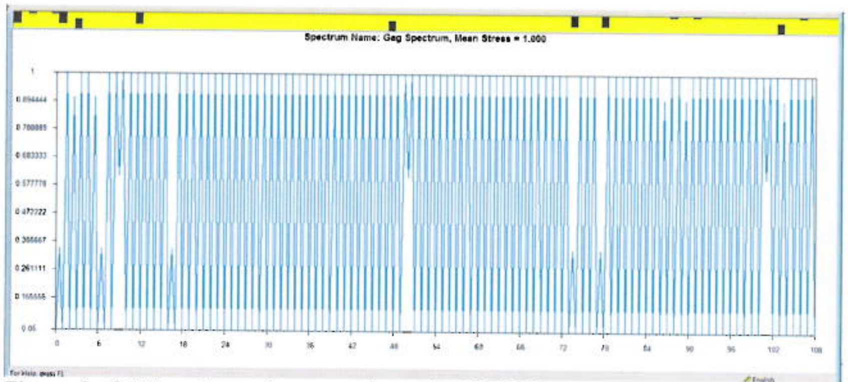

Figure 5: GAG cycle spectrum as shown in AFGROW software. This is generated from a typical mission mix provided by the OEM.

Continuous high definition (HD-1080p) video recording was done for each test using Sony NEX-VG10 camera (Sony Corporation, Japan). The recordings were reviewed after each test to determine the location of crack initiation. The high definition recording clearly shows cracks as they emanate from the fastener holes. Recorded data were archived for various post-test analyses.

The failed specimens were observed under an optical microscope to identify the regions of fatigue cracking in each panel. The data were then tabulated and a SolidWorks model built showing the location and extent of each damage site on each panel. Some of the panels had spot welded doublers. In such cases, the damage in each layer of the panel in the lap joint is characterized.

This is an on-going program. To date, 35 residual life tests have been performed. Of these, data have been reduced for 25 tests, and 14 panels have been characterized in detail for the locations and extent of damage. Overall, these panels have shown higher than expected residual life, and the damage has occurred and progressed largely as expected. The following section details the results from these tests for the 25 completed tests.

RESULTS

The panels were carefully investigated after failure to identify the locations of fatigue damage and to determine where panels failed under overload due to accumulated damage leading to fast fracture. It was easy to identify these features optically, as the fatigue regions showed planar fracture surfaces oriented predominantly perpendicular to the loading axis, while the fast fracture region was characterized by dominant shear lips. Figure 6 shows a panel immediately after failure. It can be seen that the top panel of the lap joint has failed at the topmost row of the fasteners, while the bottom panel has remained intact. The unsymmetric lap joint creates an out-of-plane load eccentricity such that the topmost row of the fasteners experiences the maximum flexural tensile load due to tension loading. This additional load makes the topmost row of fasteners more susceptible to failure. For most panels, failure occurred at this location. Some panels, however, had spot welded doublers in the lap joint, creating dual laps in one lap joint. Such panels primarily failed by the failure of the topmost row of the spot welded doubler. Again the topmost row of the spot welds experiences higher stress due to the eccentric loading and is therefore the first to fail.

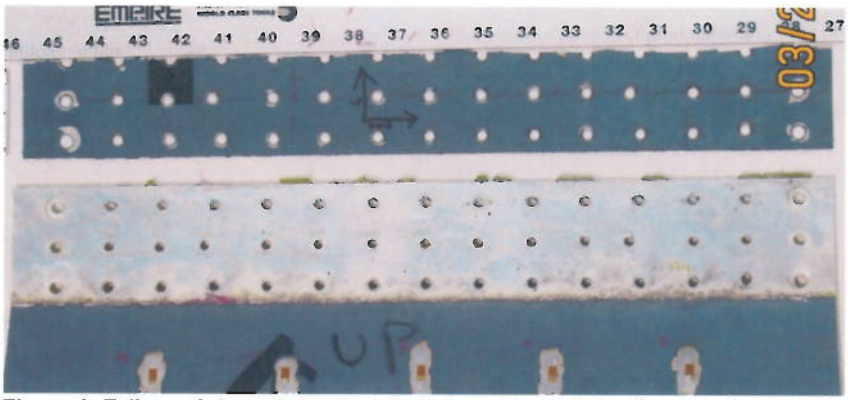

Figure 6: Failure of the test panel at the top most row of the fasteners (nearest the scale; scale in inches). It can be seen that the bottom panel is completely intact.

Table 1 lists the size of the cracks emanating from each fastener hole along the failed fastener row, shown in Figure 6, in which the 15 holes are numbered from left to right ("1" being the left-most hole). Cracks emanating from the left side of the hole are simply labelled with the hole number. Cracks emanating from the right side of the hole are labelled with a "b" suffix (e.g., 14b). The crack lengths are represented in a and c dimensions, where a is the length of the crack along the face of the skin and c is the length of the crack in the thickness direction of the panel. Where c is marked as t, it indicates that the crack has grown through the thickness of the panel. As seen from this table, fastener holes 1 and 15 showed no sign of fatigue cracking. They are the outermost rivets and, as mentioned earlier, these rivets were replaced with the next oversize rivet and were assembled with some interference specifically to avoid the crack initiation at these locations arising from edge effects. Video indicated the crack originated between fastener numbers 13 and 14. Analysis indicates the crack started from the left side of Hole 14. A few of the cracks from this panel with their identifying crack numbers are shown in Figure 7.

Table 1: Tale showing the size of the fatigue cracks emanating from each fastener hole of the failed section shown in Figure 6. Length *a* is across the panel face and *c* is the measured crack length in the thickness direction.

Fracture Surface	Fatigue?	a(mm)	c(mm)
1	No		
1b	No		
2	Yes	1.96	1.08
2b	Yes	4.42	t
3	Yes	0.72	0.16
3b	No		
4	Yes	0.578	0.357
4b	Yes	0.847	0.715
5	Yes	1.39	0.51
5b	Yes	1.44	0.833
6	Yes	3.6	t
6b	Yes	1.38	0.661
7	Yes	2.44	1.348
7b	Yes	0.927	0.517
8	Yes	6.05	t
8b	Yes	5.7	t
9	Yes	3.76	t
9b	Yes	6.8	t
10	Yes	11.07	t
10b	Yes	9.86	t
11	Yes	6.99	t
11b	Yes	5.48	t
12	Yes	5.87	t
12b	Yes	4.375	t
13	Yes	14.14	t
13b	Yes	10.62	t
14	Yes	13.25	t
14b	Yes	19.5	t
15	No		
15b	No		

Figure 7 shows the cracks emanating from fastener holes 2, 5 and 6 of the failed panel shown in Figure 6. It can be seen that this panel had multi-site damage prior to its final failure. Close examination of Figure 7 reveals that all the cracks initiated from the knife edge of the countersink hole. This knife edge, due to its sharpness, has a higher stress intensity factor. Note also that the bottom of the countersink is in close proximity to the bottom panel of the lap joint. The actual initiation site could be due to fretting with the bottom panel, but detailed microstructural examination is beyond the scope of the present work. Data have been compiled for 14 panels in similar fashion. Furthermore, we have also created SolidWorks models for each panel showing the location and the extent of each crack. Figure 8 shows the SolidWorks model where the cracks are superimposed at the fastener hole locations.

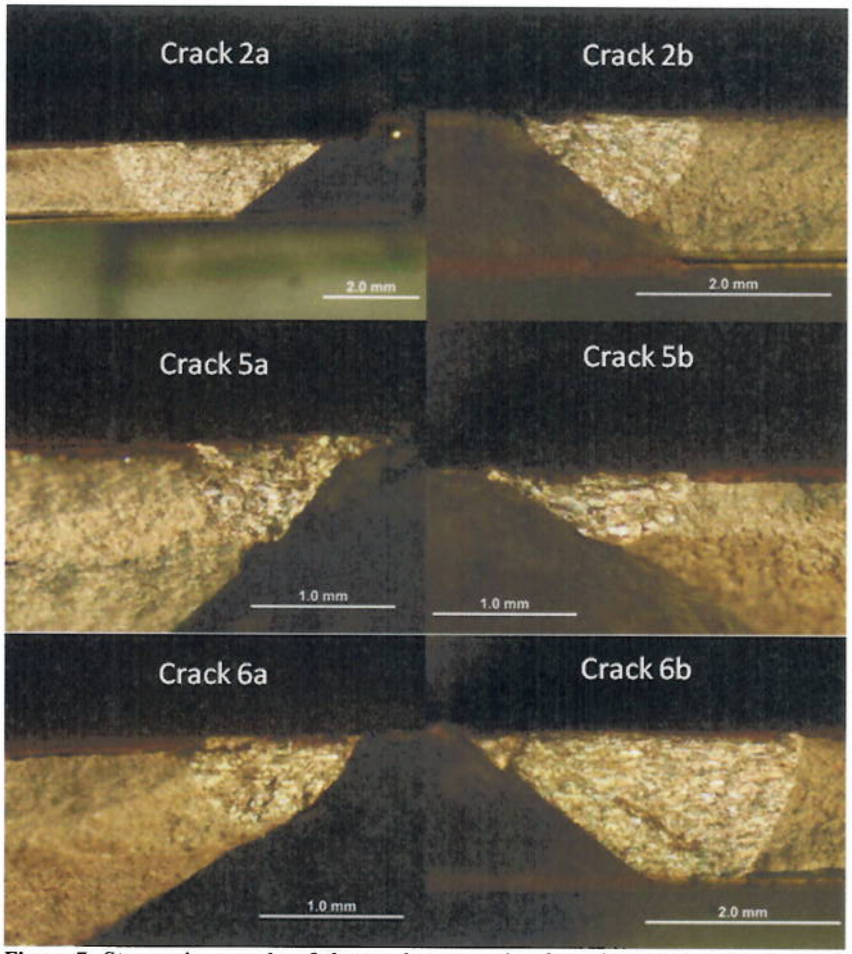

Figure 7: Stereomicrographs of the cracks emanating from fastener holes 2, 5 and 6. The shiny surfaces are the fatigue cracks and the dull surfaces are the fast fracture regions.

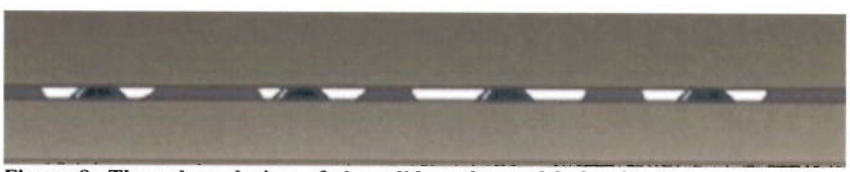

Figure 8: The enlarged view of the solid works model showing the cracks at hole locations 8, 9, 10 and 11 only. This model can be adapted to most any FEA software.

In some instances, doublers were found to have been used in the lap joint configuration. In such cases, the failure occurs in both panels and crack lengths are different from those seen in the welded panels. Figure 9 shows the failure of the panel which had a spot-welded doubler on the top panel. While the failure occurred at the fastener row of the lap joint, it can be seen that the crack propagates quite differently in both panels. The crack does not initiate from a common point of two panels, even though fretting is observed there. The cracks in this case are emanating from the bottoms of both of the panel skins. The bottom edge has a knife edge due to the countersinking operation. These knife edges are the initiation sites for both of the panel skins. Also, the lengths to which the cracks grow are different for the two panels. This is truly a case of multi-site damage (MSD) and multi-element damage (MED) which, when taken together, constitute wide-spread fatigue damage (WFD). It can be seen that this change in configuration has taken the simple case of MSD in one row of fasteners (previous case) to a much more complex one having the effects of inter-panel fretting and differential knife edges from a single countersinking operation. The fretting also indicates the presence of burrs from the countersinking operation. The panels were first spot welded and then machined to create the countersink for the fasteners. This machining operation appears to have left burrs and allowed some chips to become lodged between the two panels, leading to fretting. Modelling details all of these scenarios would be a very daunting task. The aim of the present work is therefore to create the database of panel failures and their crack locations and dimensions. The data are compiled in a similar manner as described above for all 14 test panels and are available to use for any kind of modelling. Detailed knowledge of the stress spectrum would also be required.

Table 2 shows the cycles to failure for each panel. These cycles to failure are actual GAG cycles after removing the marker band cycles and normalizing for the mission mix.

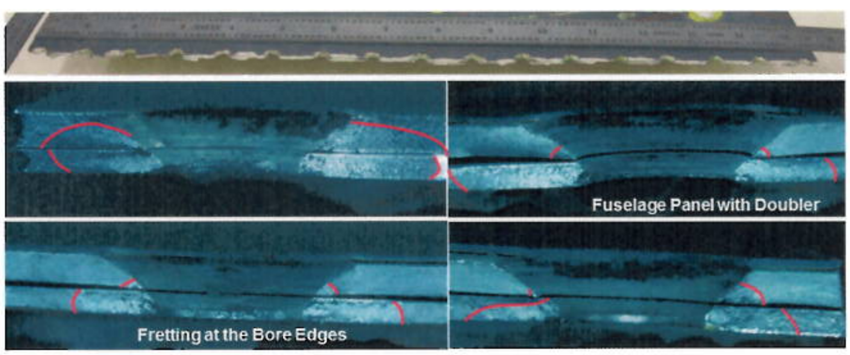

Figure 9: Failure of fuselage panels with spot welded doublers. It can be seen that the damage progresses in the two panels quite differently. In both panels the crack initiates from the bottom, where the countersink forms a knife edge.

Table 2: Table showing the residual life of 25 fuselage lap joint panels.

Name	Width (in.)	Thicknes s (in.)	Thicknes s (in.)	# of Rivets in Joint	Rivet Diameter (in.)	# cycles to failure
TS 604	16	0.043	0.046	51	0.1875	154984
TS 612A	16	0.062	0.051	36	0.1875	67057
TS 612F	16	0.062	0.051	36	0.1875	96145
TS103L	16	0.074	0.060	36	0.208	138889
TS103C	16	0.048	0.120	34	0.1875	350000
TS628	16	0.095	0.125	51	0.1875	40619
TS623	16	0.055	0.074	33	0.1875	114196
TS625	17	0.057	0.074	33	0.1875	134413
TS606	16	0.095	0.077	48	0.1875	308870
TS609	16	0.042	0.043	31	0.1875	341750
TS612	21.5	0.060	0.050	45	0.1875	107958
TS616	14.5	0.062	0.056	39	0.2188	158598
TS634	19	0.040	0.042	36	0.2188	59626
TS637	19	0.040	0.042	36	0.2188	109073
TS622	11.5	0.061	0.044	33	0.2188	235163
TS635	18	0.049	0.062	39	0.2188	165769
TS623	17.25	0.048	0.064	44	0.2188	92649
TS621	18.5	0.106	0.098	54	0.2188	798776
TS611	19.5	0.060	0.060	48	0.1875	95760
TS607	18.5	0.070	0.080	54	0.1875	193294
TS608	14.5	0.060	0.061	36	0.2500	45865
TS610	18.5	0.083	0.080	54	0.1875	265632
TS608	18.5	0.085	0.080	54	0.1875	246144
TS604	19	0.043	0.056	39	0.2500	145041
TS602	19.8	0.075	0.072	57	0.2188	98322

Apart from the residual life of the lap joints, Table 2 also lists the width and the thickness of the two panels comprising the lap joint, as well as the number and size of the fasteners used to fabricate the joint. Regarding the number of cycles to failure, it should be noted that post-test examination showed none of the panels had any pre-existing (in-service) damage at any of the fastener holes. This suggests that the crack length data collected are only from the crack growth that has occurred during testing. Because these panels have seen many in-service pressurization cycles, only the *residual lives* of these panels are being measured here. To actually calculate the total number of cycles to failure (including time for nucleating cracks) would require a detailed service history of the aircraft. Nonetheless, these data can be used effectively for predicting remaining life of other aircraft in same fleet.

Table 2 also provides an interesting insight into the behaviour of test panels under spectrum loading. For example, TS 606 and TS 609 have the same width, but TS 606 has much thicker skin panels as compared to TS 609. However, TS 609 shows a higher number of cycles to failure: 341,750 as compared to 308,870. Since the loading per inch of panel width is based on the internal pressure, the actual load experienced by both panels is the same, since their width is the same. But the resulting hoop stress is almost double in the case of TS 609, which has half the skin thickness. Even so, the residual life is higher for TS 609. The reason is thought to be the fewer number of fasteners in the TS 609 joint, which reduces the overall stress intensity and thereby increases the life of the lap joint. Similarly, other panels show non-intuitive results (some admittedly due to scatter). Furthermore, there is huge variability in the life of the lap joints, from about 40,000 cycles to 800,000 cycles. However, all tested panels have exceeded the minimum expected residual life for these aircraft; the joints exhibit excellent performance even after the retirement of the aircraft.

FUTURE WORK

Analysis continues on the remaining sections that have undergone residual life testing. About 40 panels are to be eventually tested in this program.

These data will be used for Monte-Carlo simulation of MSD failures in aging aircraft structure. The pseudo-random Monte Carlo simulation based on the data generated from this study may provide better predictions of life for fuselage lap joints in similar aircraft structures.

Additionally, more thorough and advanced failure analyses may be performed on these panels to try to identify the actual number of cycles associated with each crack length to generate a database of crack growth rate per cycle (da/dN). Along with these data, beta factors can be calculated for the specimen geometry to get accurate ΔK values and generate table look-ups for AFGROW or NASGRO programs. These would be made available to multiple users for deterministic modelling of the fuselage structures with known geometry, material properties and physical condition.

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